

Final Report

A Study of System Requirements for Phobos/ Deimos Missions

Volume I Summary

June 1972

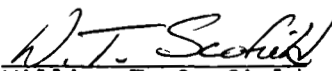


Contract NAS1-10873

A STUDY OF SYSTEMS REQUIREMENTS
FOR PHOBOS/DEIMOS MISSIONS
FINAL REPORT

Volume I
Summary

Approved


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FOREWORD

This is Volume I of the Final Report on A Study of Systems Requirements for Phobos/Deimos Missions, conducted by the Martin Marietta Corporation.

This study was performed for the Langley Research Center, NASA, under Contract NAS1-10873, and was conducted during the period 4 June 1971 to 4 June 1972. Mr. Edwin F. Harrison of Langley Research Center, NASA, was the Technical Representative of the Contracting Officer. The study was jointly sponsored by the Advanced Concepts and Mission Division of the Office of Aeronautics and Space Technology (OAST) and the Planetary Programs Division of the Office of Space Sciences (OSS) in NASA Headquarters.

This Final Report, which summarizes the results and conclusions of the three-phase study, consists of four volumes as follows:

- Volume I - Summary
- Volume II - Phase I Results - Satellite
 Rendezvous and Landing Missions
- Volume III - Phase II Results - Satellite Sample
 Return Missions and Satellite Mobility
 Concepts
- Volume IV - Phase III Results - Combined Missions
 to Mars and Its Satellites

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ABBREVIATIONS AND SYMBOLS

a	orbit semi-major axis
ACS	attitude control system
ARU	attitude reference system
Ax, Ay, Az	body acceleration
Az	azimuth angle
bps	bits per second
CC&S	control computer and sequencer
cg	center of gravity
db	decibel
DLA	declination of launch asymptote
DSN	Deep Space Net
DSS	Deep Space System
EL	elevation angle
ETC	engineering test capsule
$F1_c, F2_c, F3_c, FN_c$	lander engine thrust command
FOV	field of view
g	acceleration due to gravity, Earth
G&C	guidance and control
GCSC	guidance, control and sequencing computer
grms	gravity (rms)
HZ	hertz
i	orbit inclination
IRU	inertial reference unit
JPL	Jet Propulsion Laboratory
kbps	kilobits per second
km	kilometers
L/E	launch/encounter

LOS	line-of-sight
LPCA	lander pyrotechnic control assembly
LRC	Langley Research Center
mbps	megabits per second
MCC	midcourse correction
MLI	multilayer insulation
MMC	Martin Marietta Corporation
MOI	Mars orbit insertion
NASA	National Aeronautics and Space Administration
NW	net load factor times weight
OSR	optical solar reflector
p, q, r	body attitude rates
PTC	proof test capsule
PTO	proof test orbiter
R	range
\dot{R}	range rate
RCS	reaction control system
RF	radio frequency
RSS	root-sum-of-squares
RTG	radioisotope thermoelectric generator
R99	99 percentile closest approach radius
S/C	spacecraft
TA	orbit true anomaly
TEI	trans-Earth injection
T/M	thrust-to-mass
TMI	trans-Mars injection
TWTA	traveling wave tube amplifier
UHF	ultra-high frequency
UV	ultraviolet
VHE	hyperbolic excess velocity

VM	velocity meter
VO	Viking Orbiter
VRU	velocity reference unit
W	weight
α	solar absorptivity
ΔV	delta velocity
ΔV_{STAT}	navigation uncertainty delta velocity
ϵ	orbit eccentricity
θ	pitch attitude angle
π	3.1416
ρ	density
σ	standard deviation
ϕ	roll attitude angle
ψ	yaw attitude angle
μ_{MARS}	Mars central gravity potential constant
Ω	Mars longitude of ascending node
ω	Mars argument of periapsis
\sim	approximately

I. WHY EXPLORE THE MOONS OF MARS?

Phobos and Deimos, the two natural satellites of Mars, hold unique places in the solar system. The smallest of the 31 known satellites, and exhibiting extremely low albedos, they are thought by some to be captured asteroids rather than remnants of the Mars accretion process. If they are asteroids, composed of primordial matter from the original solar nebula, they may hold clues vital to our understanding of the origin and evolution of our solar family. If, on the other hand, Phobos and Deimos were derived from their mother planet and have since been preserved in an unmodified state, they may yield valuable insight into the history and evolution of Mars in particular and planets in general. But still other theories of origin have been offered: that the satellites are fragments of a larger synchronous companion of Mars that was destroyed in a cataclysmic collision; and, that Phobos and Deimos are of separate origins, representing both indigenous and alien material.

In spite of, or perhaps more likely, because of the mystery and disagreement that surrounds the Martian satellites, they have stimulated considerably scientific interest. Therefore, space flight missions to observe and analyze these small bodies promise rich scientific returns, closely attuned to our Nation's stated objectives for planetary exploration.

II. WHAT DO WE KNOW ABOUT PHOBOS AND DEIMOS?

Phobos and Deimos were first discovered by Asaph Hall in 1877 using a 66 cm refractor at the Washington Observatory. Since that time, they have been observed by telescope, caught fleetingly by the camera of the Mariner 7 flyby spacecraft, and most recently photographed by the television imaging system of Mariner 9. The Mariner 9 observations*, the best of which were at a range of about 5500 km, provided feature resolution on the surface of Phobos of approximately 450 meters. Figure II-1 is a Mariner 9 photo of Phobos showing the heavily cratered surface, and Figure II-2 is one of Deimos taken at longer range. From the Deimos image, one might infer a somewhat smoother surface.

In addition to the visual characteristics shown in these photos, practically all the known physical data on the Martian satellites are summarized in the following table.

Table II-1 Characteristics of the Satellites of Mars

PROPERTY	PHOBOS	DEIMOS
SEMI-MAJOR AXIS OF ORBIT, km	9382	23480
ECCENTRICITY	0.0170	0.0028
INCLINATION OF ORBIT TO MARS EQUATOR, deg	0.95	1.3
ORBITAL PERIOD, hr	7.654	30.298
ROTATIONAL PERIOD, hr	7.654	?
SIZE, km	21 x 25	12 x 13.5
ALBEDO	.06	.05
SURFACE GRAVITY, EARTH G	.001	.0005

* As predicted by Edwin F. Harrison and Janet W. Campbell, NASA Langley Research Center: "Reconnaissance of Mars Satellites." *J. of Spacecraft and Rockets*, Vol. 7, No. 10, October 1970.

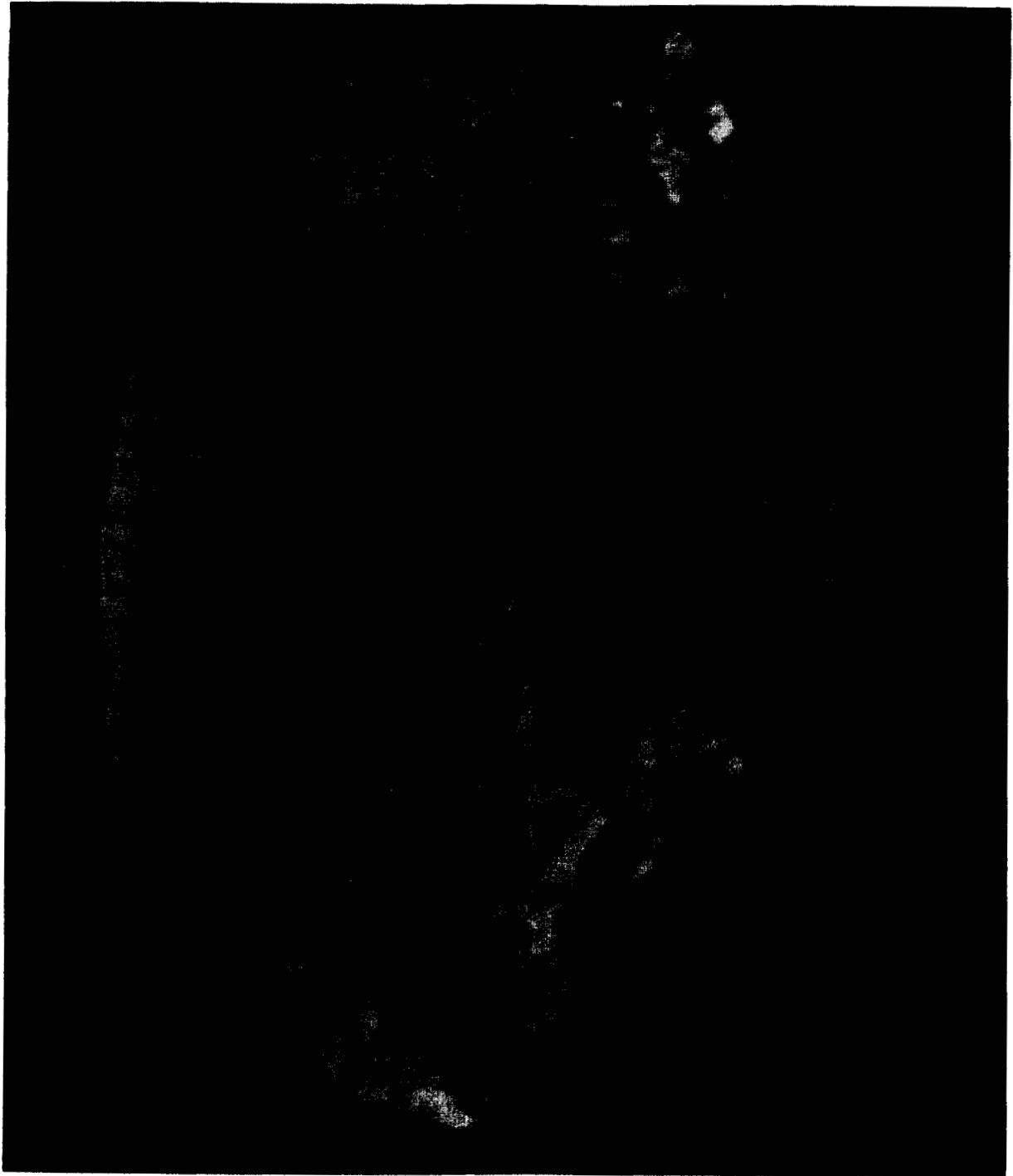


Figure II-1 Mariner 9 Photo of Phobos

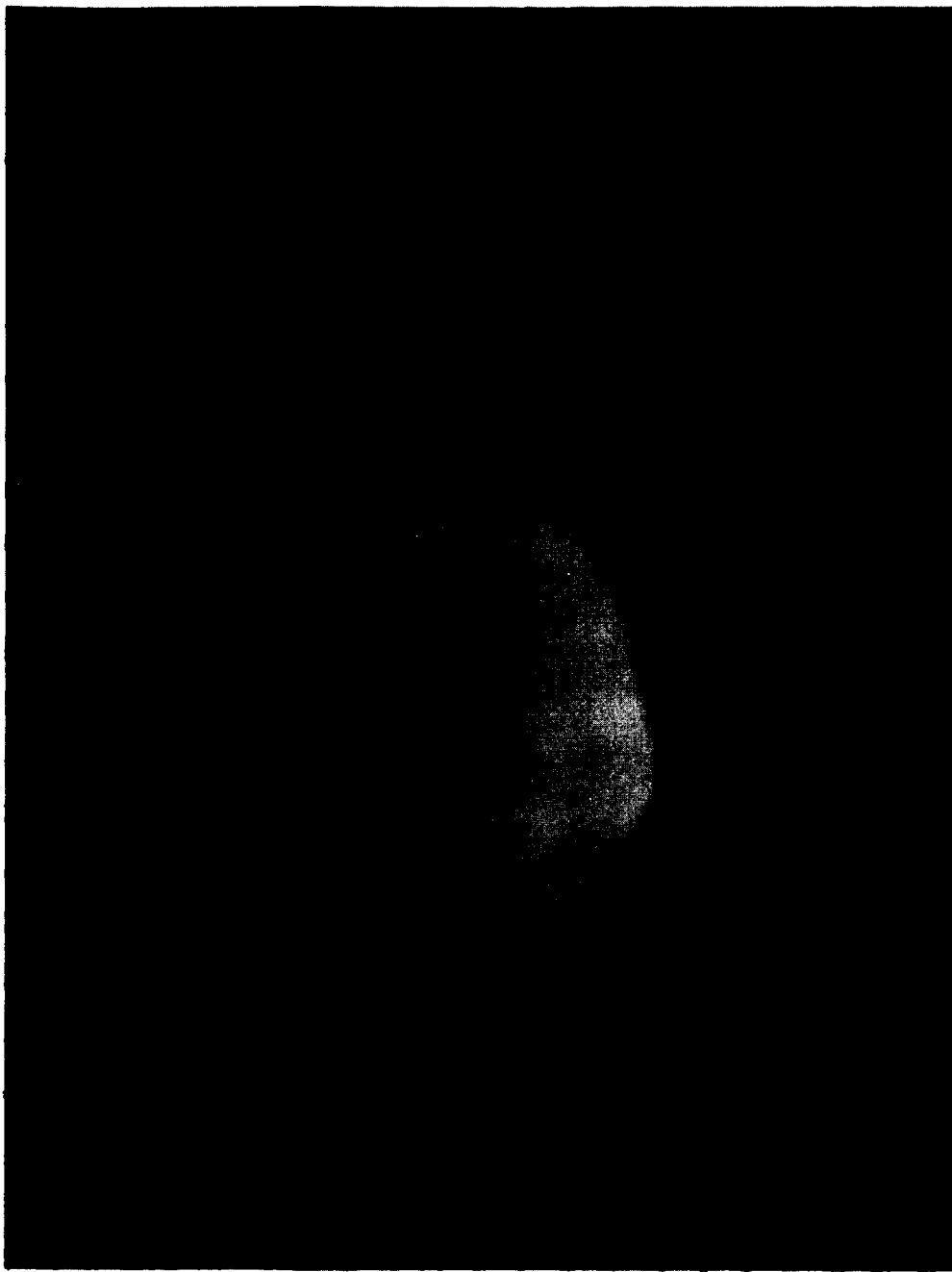


Figure II-2 Deimos as Seen by Mariner 9

III. SCIENTIFIC OBJECTIVES OF PHOBOS/DEIMOS EXPLORATION

The scientific objectives for a planetary exploration mission can be most often expressed as a series of questions. After talking to a number of members of the scientific community and reviewing literature on the Martian satellites, the questions summarized in Table III-1 emerged as the scientific rationale for exploring Phobos and Deimos.

In order to provide answers to such questions, specific scientific observations and measurements must be made. Table III-2 outlines the science investigations and associated experimental and instrumental techniques that should be considered for missions to the vicinity of the satellites.

The tabulation shown here is, of course, not complete, but illustrates the kinds of experiments that have been performed in previous programs such as Apollo, Surveyor, and Viking, to determine the composition, origin, and evolution of planetary bodies. In carrying out useful and practical space missions, science teams, NASA management, and contractors must work together to select instrument payloads that will return high priority data for minimum cost investment. Decisions like whether age dating should be attempted in situ or whether it justifies the performance of sample return missions, are typical of the mission science planning process.

Table III-1 Phobos/Deimos Science Questions

Origin:	
	Remnants of Mars Accretion?
	Remnants of Larger Mars Companion?
	Captured Asteroids?
Evolution:	
	Age?
	Impact History?
	Weathering and Modification?
	Reheating?
Comparative Planetology:	
	Earth/Moon Correlations?
	Phobos/Deimos Differences?
	Clues to Understanding Mars Evolution?
	Observation of Primordial Material?

Table III-2 Science Objectives for Missions to Phobos and Deimos

INVESTIGATION	TECHNIQUE
<u>Dynamical Characteristics</u>	
Orbital Elements	Imagery
	Tracking Transponder
Rotational Motions, Librations	Imagery
<u>Physical & Chemical Characteristics</u>	
Mass	Perturbations to Spacecraft Motions
Size, Shape & Volume	Imagery
Bulk Density	From Mass, Size, Shape & Volume
Surface Morphology (Survey)	Imagery (1-m Resolution)
Surface Morphology (Local Features of Interest)	Imagery (10-cm Resolution)
Gross Surface Composition & Homogeneity (Survey)	Spectral Reflectivity (Visible, Near & Mid-IR)
	X-Ray Fluorescence

Table III-2 Science Objectives for Missions to Phobos and Deimos (Concluded)

INVESTIGATION	TECHNIQUE
Surface Composition (Local Surface & Drill Core Sample)	
Major & Minor Elements	Alpha Backscattering
	X-Ray Fluorescence Spectrometry
Trace Elements	Neutron Activation
Radioactive Elements (U, Th, K)	Gamma Ray Spectrometry
	Passive Alpha Spectrometry
Mineralogy	X-Ray Diffraction
	Quasi-Microscopy
Water of Crystallization, Trapped Gases	Thermal Release/Mass Spectrometry
Organic Compounds	Pyrolyzer/Mass Spectrometry
Density of Surface Materials	Gamma Ray Backscattering
Internal Density & Homogeneity	Active Seismometry
Crystallization Age	Coarse Age Dating

IV. BACKGROUND AND APPROACH FOR THIS STUDY

The original work to define space flight missions to Phobos and Deimos was done at the NASA Langley Research Center by E. Brian Pritchard and Edwin F. Harrison*. Pritchard and Harrison studied the performance requirements of a number of mission modes and concluded that Phobos/Deimos exploration was feasible using current space vehicle concepts and technology.

In June of 1971, Martin Marietta Corporation's Denver Division was awarded Contract NAS1-10873, a study of Systems Requirements for Phobos/Deimos missions. Our direction was to start with the Pritchard/Harrison work and proceed through the following general steps:

1. Assess scientific interest in Phobos and Deimos;
2. Develop mission design(s);
3. Define practical systems and subsystems;
4. Estimate costs and schedules;
5. Recommend baseline program(s);
6. Identify technology requirements.

Overall guidelines provided by the NASA Technical Representative of the Contracting Officer emphasized the following:

1. Design missions around realistic science objectives and payloads;
2. Minimize program costs;
3. Emphasize 1979 and 1981 opportunities;
4. Apply proven spacecraft hardware (Mariner, Viking, Pioneer);
5. Consider expendable and reusable launch systems.

* See *Phobos/Deimos Missions*, AIAA Paper No. 71-830, presented at the AIAA Space Systems Meeting at Denver, Colorado, July 19-20, 1971.

The first two guidelines were considered most important. The study was conducted under the assumption that a Mars satellite mission would be approved only if a high value could be established and recognized for the potential scientific return, and at the same time, only if the program costs could be held to a range competitive with other desired new space flight starts.

Our approach to this work was to examine a series of baseline mission concepts. In Phase I of the study, a basic Phobos or Deimos rendezvous and landing mission was defined. In Phase II, using the understanding of the landing mission developed in Phase I, a baseline satellite sample return mission was derived. Finally, in Phase III, a large number of combination Mars and Phobos/Deimos exploration missions was examined to select a recommended baseline combination profile that would offer high science return value for minimum cost. The results of these three study phases are described in detail in Volumes II, III, and IV of this report and are summarized in the remaining pages of this volume.

Phase I - Satellite Rendezvous, Landing and Roving Mission

V. PHASE I - SATELLITE RENDEZVOUS, LANDING AND ROVING MISSION

It was decided that the initial three months of the study should be spent in compiling basic information needed for any satellite mission design and in defining a baseline rendezvous and landing mission upon which other mission modes could be built.

The first task was to develop a Phobos/Deimos Engineering Model, patterned after the Mars Engineering Model used in the Viking Project. The model, included in Appendix A of Volume II of this report defines the Phobos/Deimos surface characteristics and environments to which a spacecraft would have to be designed. Some of the more important satellite features assumed in the model are:

1. Rough topography due to saturation cratering quite likely;
2. Surface composition could be:
 - a. Martian-like (Basaltic)
 - b. Meteorite-like
 - (1) stones (rock)
 - (2) irons (metallic, predominantly Fe and Ni)
 - (3) carbonaceous chondrites (high in organics, very friable);
3. Soil bearing capacity could vary from extremely poor (loose rubble) to excellent (bare rock or iron);
4. Dust layer assumed to be one centimeter thick;
5. Moisture free surface and negligible gravitational forces will enhance electrostatic effects on dust;
6. Synchronous rotation (spin period equals orbital period) assumed.

The next task was to define a realistic science payload for an initial rendezvous and landing mission. Table V-1 outlines the science objectives and recommended instrumentation for the baseline payload. The total weight of these instruments, including both orbital and landed science, would be approximately 90 kg.

Table V-1 Phobos/Deimos Science Objectives - First Mission

	<u>Science Objective</u>	<u>Candidate Instrumentation</u>
Orbiter	Mineralogy	Visible and IR Spectrometry
	Elemental Composition	X-Ray Fluorescence (Solar-Stimulated)
	Radioactive Elements	Gamma-Ray Spectrometer
	Imagery	Television Camera
Lander	Mineralogy	X-Ray Diffraction; Optical Microscopy
	Elemental Composition	Alpha Backscatter; X-Ray Fluorescence
	Radioactive Elements	Gamma-Ray Spectrometer
	Imagery	Facsimile Camera

A. RENDEZVOUS AND LANDING MISSION PROFILE

The basic event sequence used to rendezvous with the satellites of Mars is shown in Figure V-1. The observation orbit (step 7) takes advantage of the fact that the orbital periods of Deimos and Phobos are very close to a ratio of 4 to 1 (approximately 30 and 7.5 hours). The observation orbit, with a period of 15 hours, will allow close encounter with Deimos every other revolution and repeated close crossings of Phobos. A decision to rendezvous with either satellite could be made after analyzing data gathered in the observation orbit. To rendezvous with Deimos, the observation orbit is circularized with a burn at apoapsis. To go to Phobos, the periapsis of the observation orbit must be raised from 2660 km altitude to the 6000 Km altitude of the Phobos orbit and circularized there.

The delta velocity requirements for the major maneuvers in this sequence are:

Mars orbit insertion	980 mps (1979 launch)
Plane change	65 mps
Transfer to observation orbit	300 mps
Raise periapse to Phobos	200 mps
Circularize at Phobos	430 mps
Rendezvous and landing	50 mps

Adding in other miscellaneous velocity requirements brings the total budget for the Phobos mission (after the initial insertion from Earth orbit to the trans-Mars trajectory) to 2375 mps. A complete Monte Carlo error analysis simulation was developed and run for this mission sequence to prove that a rendezvous with Phobos could be achieved 99% of the time within sufficiently

1. Mars Orbit Insertion

2. Coast In Capture Ellipse

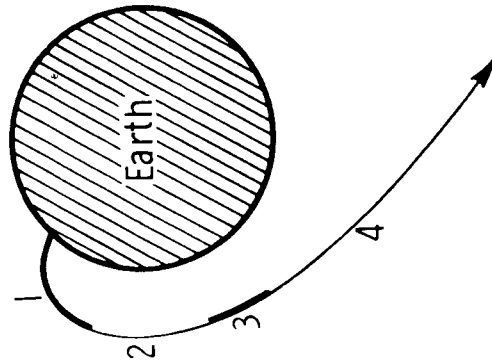
3. Plane Change Maneuver

4. Capture-To-Phasing Maneuver

5. Phasing Orbit

6. Phasing-To- Observation Maneuver

7. Observation Orbit

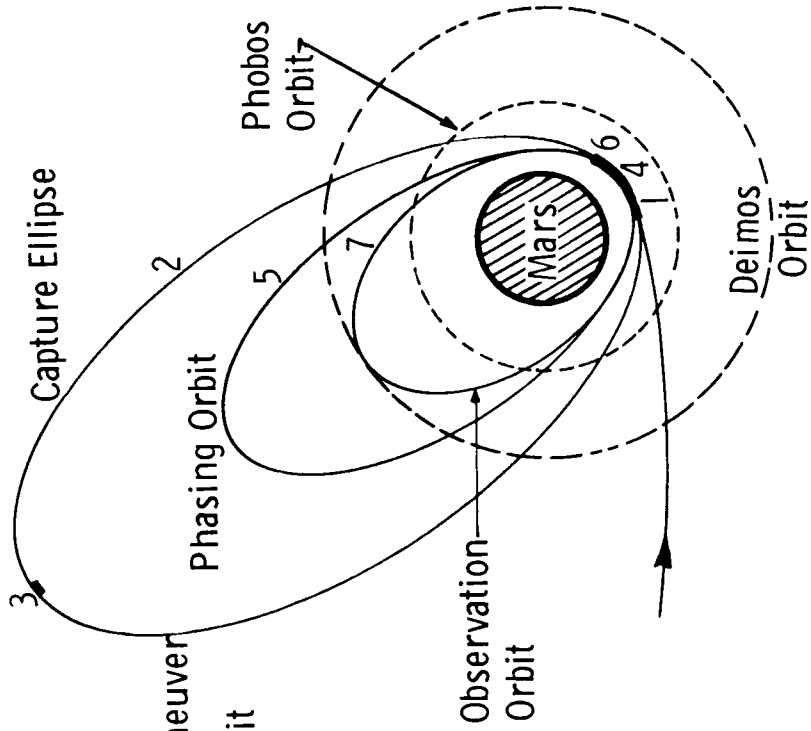


1. Launch to Circular Park

2. Coast In Circular Park

3. Trans Mars Injection

4. Trans Mars Coast



Not To Scale

Figure V-1 Overview of Mission Profile

close range to permit final closure under the control of a rendezvous radar. The error sources considered in this simulation were:

- Mars orbit insertion uncertainties;
- DSN tracking uncertainties;
- Satellite ephemeris uncertainties;
- Spacecraft execution errors;
- TV camera pointing errors.

B. RENDEZVOUS AND LANDING MISSION BASELINE SPACECRAFT

Figure V-2 depicts the baseline spacecraft recommended as the result of our Phase I study. After being launched, inserted into Earth parking orbit and then injected into the trans-Mars trajectory by a Titan IIIE/Centaur launch vehicle, the modified Viking '75 Orbiter shown here, delivers the payload to the Martian satellite. To perform the orbital **maneuvers at Mars shown in the mission** profile, the Orbiter propulsion system must be "stretched" in capacity by 38%. This is done by increasing the sizes of the main propellant tanks approximately 13cm in both length and diameter. The pressurization sphere must also be enlarged by 8 cm in diameter. The total injected weight of the baseline spacecraft is approximately 3600 Kg (7900 lbs) which is close to the injected weight of the Viking '75 spacecraft.

In this baseline, the landed science payload is delivered to the satellite surface with a separable lander vehicle shown above the Orbiter. The lander shown in Figure V-3 derives its design from the Viking '75 Lander although significant changes have been made to accommodate the landing technique and the gravity field associated with the satellites. The lander shown weighs 482 Kg (1063 lbs) including an 82 Kg allocation for science payload.

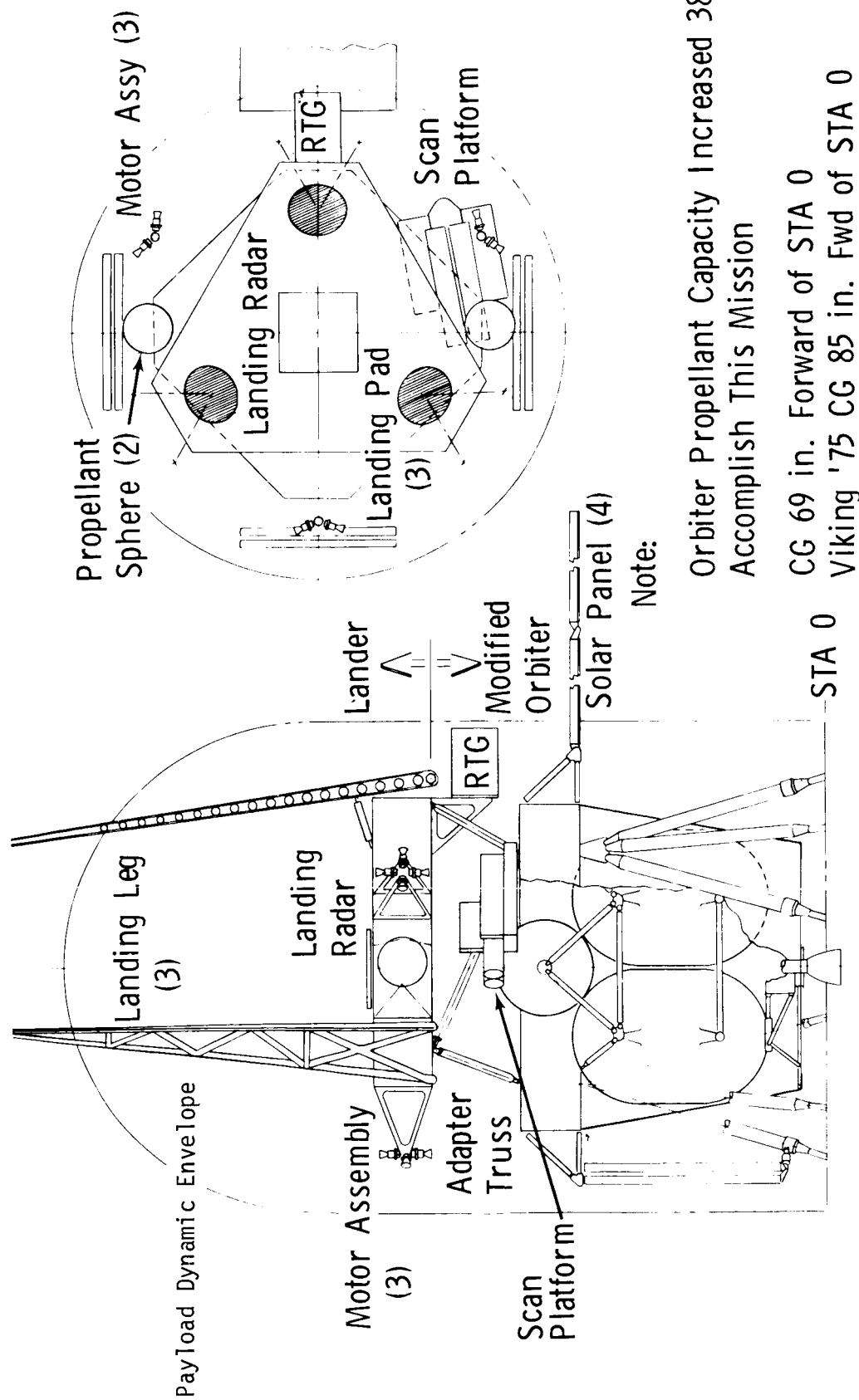


Figure V-2 Phobos/Deimos Lander/Rover (Baseline Configuration)

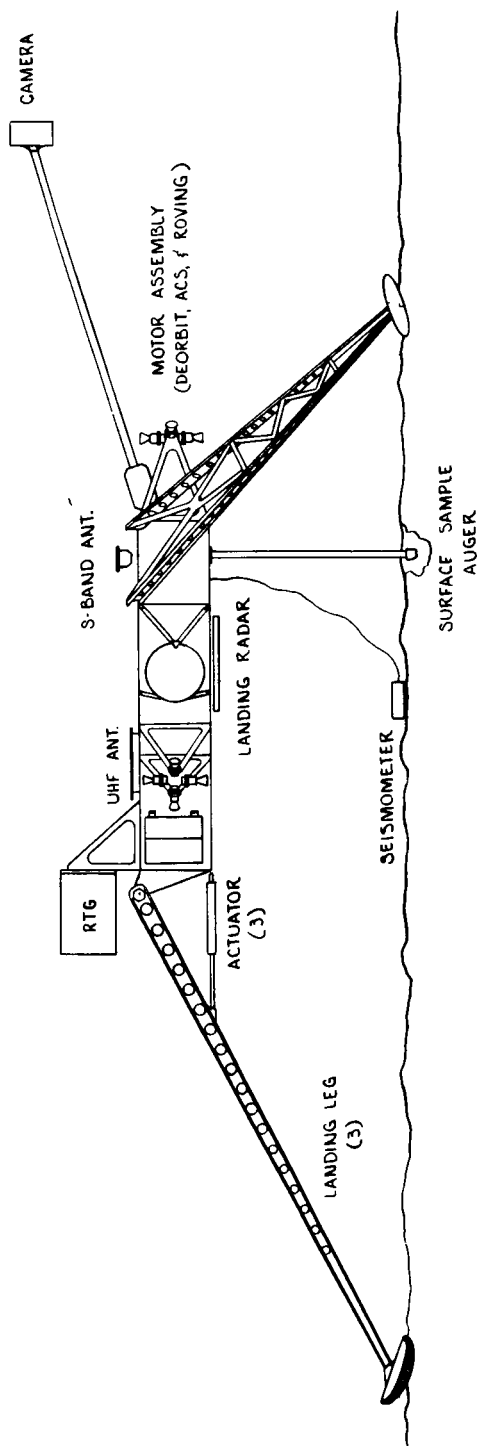


Figure V-3 Phobos/Deimos Lander/Rover

The recommended landing sequence beginning with the orbiter and lander in a station keeping co-orbit with the satellite is depicted in Figure V-4. A rendezvous radar, derived from a modified Viking '75 Lander radar altimeter and having a range of 100 km, is mounted on the lander and, after locking on the satellite, provides line of sight (LOS), range, and range rate information for the rendezvous maneuvers. The rendezvous control scheme is the same as used in the Apollo and Gemini programs and by the Russians in their automatic rendezvous of space station elements. Propulsive thrust control for the lander after separation for the Orbiter is provided by Viking '75 Lander attitude control thrusters. The same thrusters provide propulsive "hopping" surface mobility to the lander configuration after landing.

An alternate landed mobility technique was analyzed in Phase II of this study that employed wheels instead of propulsive hopping. Because of the low gravity on Phobos (the baseline lander would weigh an equivalent to 1 Earth pound on the surface) very light wheel construction can be used as shown in Figure V-5. A total of 12 Kg is added to the lander to provide wheels and drive assemblies. In trade studies, the wheeled version turned out to be lighter and more mobile than the flying lander.

As an alternative to the basic idea of an orbiter spacecraft and a separable satellite lander, a landed orbiter configuration was defined as shown in Figure V-6. By landing the entire orbiter, greater landed payloads and more efficient use of supporting subsystems can be achieved. In this case, the landed weight allocation to science or other payload can be increased from 82Kg to about 500Kg. Adapting an interplanetary cruise vehicle like the Viking Orbiter to a lander role does involve more extensive modifications however.

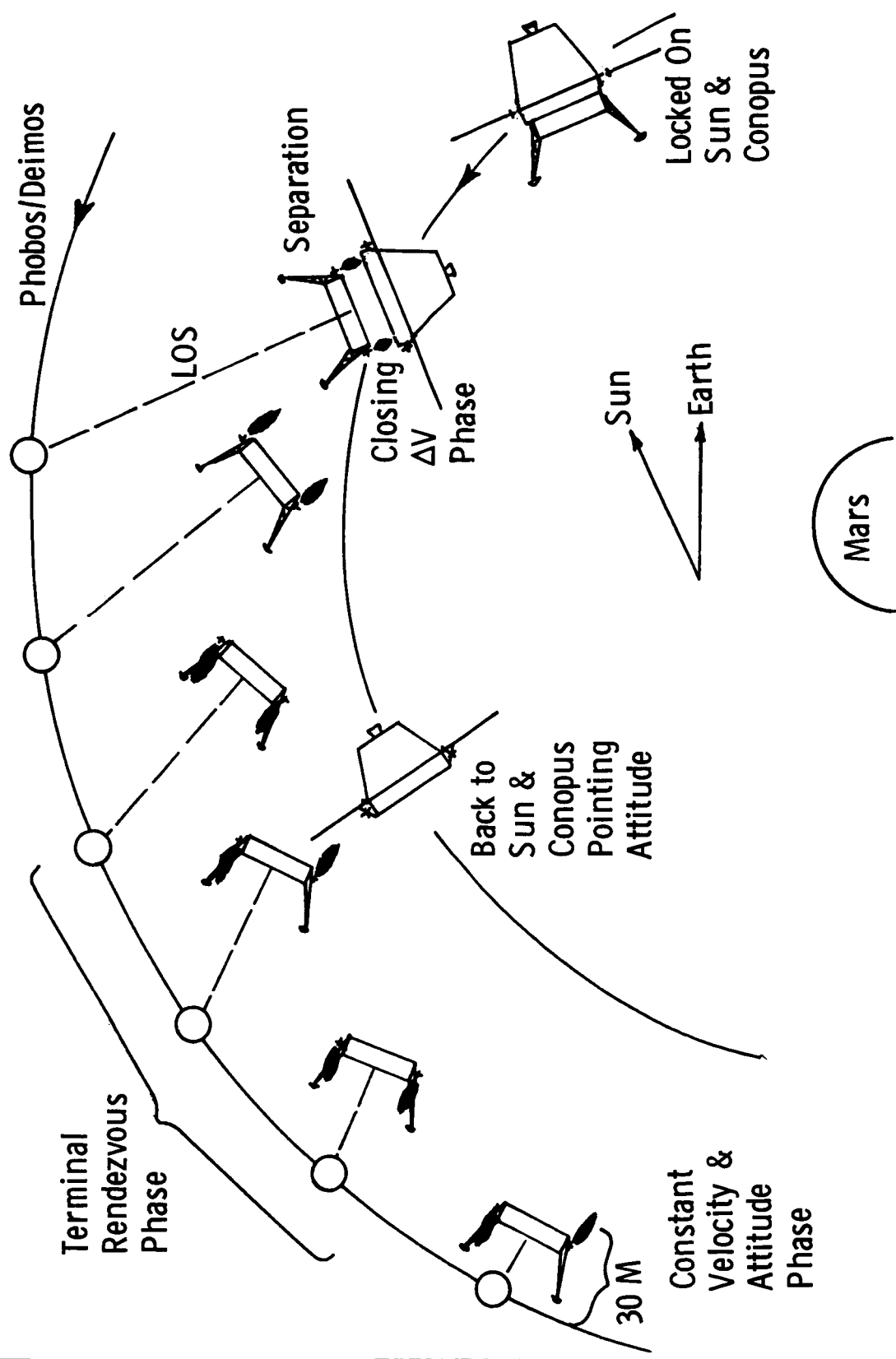


Figure V-4 Satellite Landing Sequence

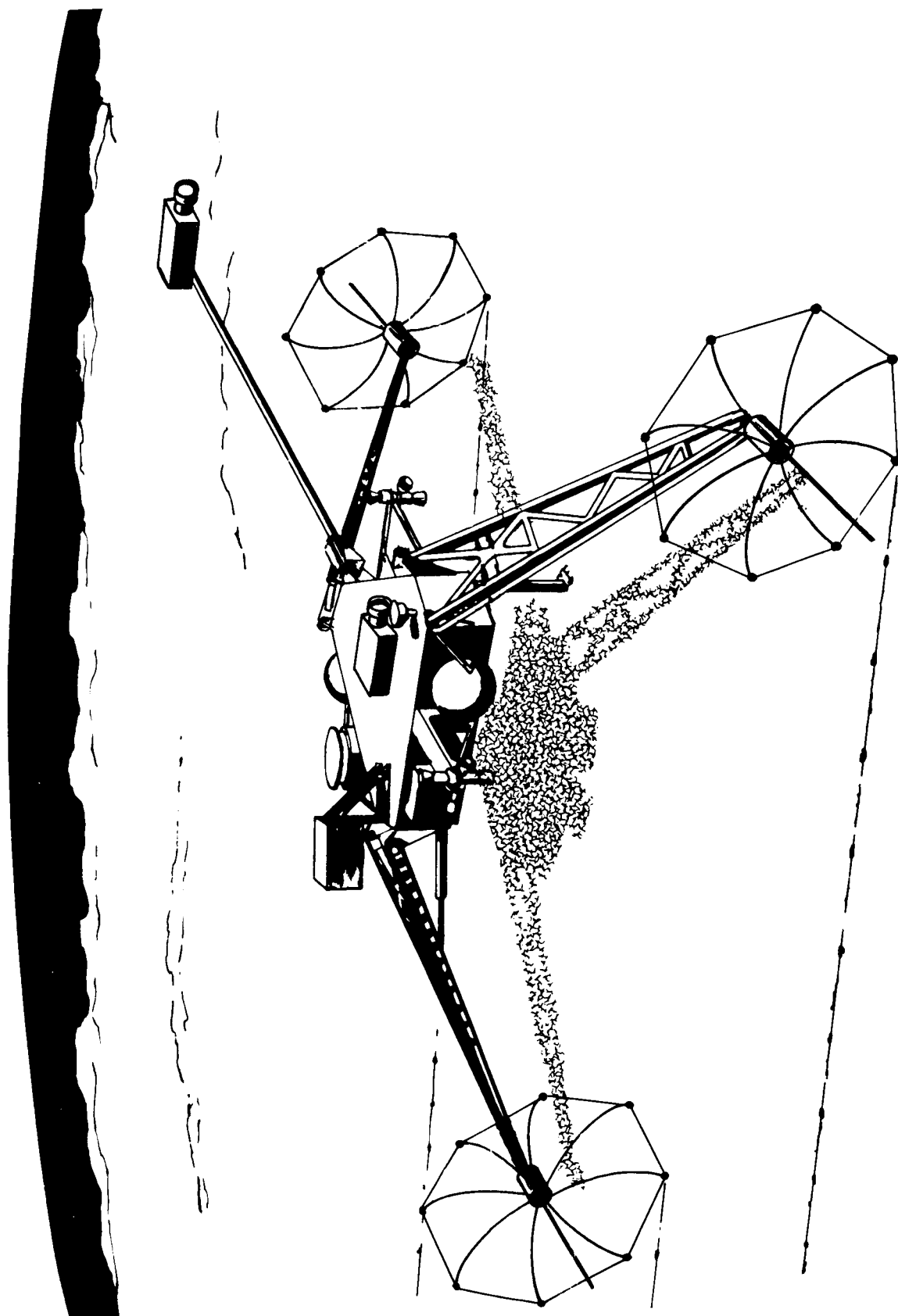


Figure V-5 Phobos/Deimos Wheeled Roving Lander

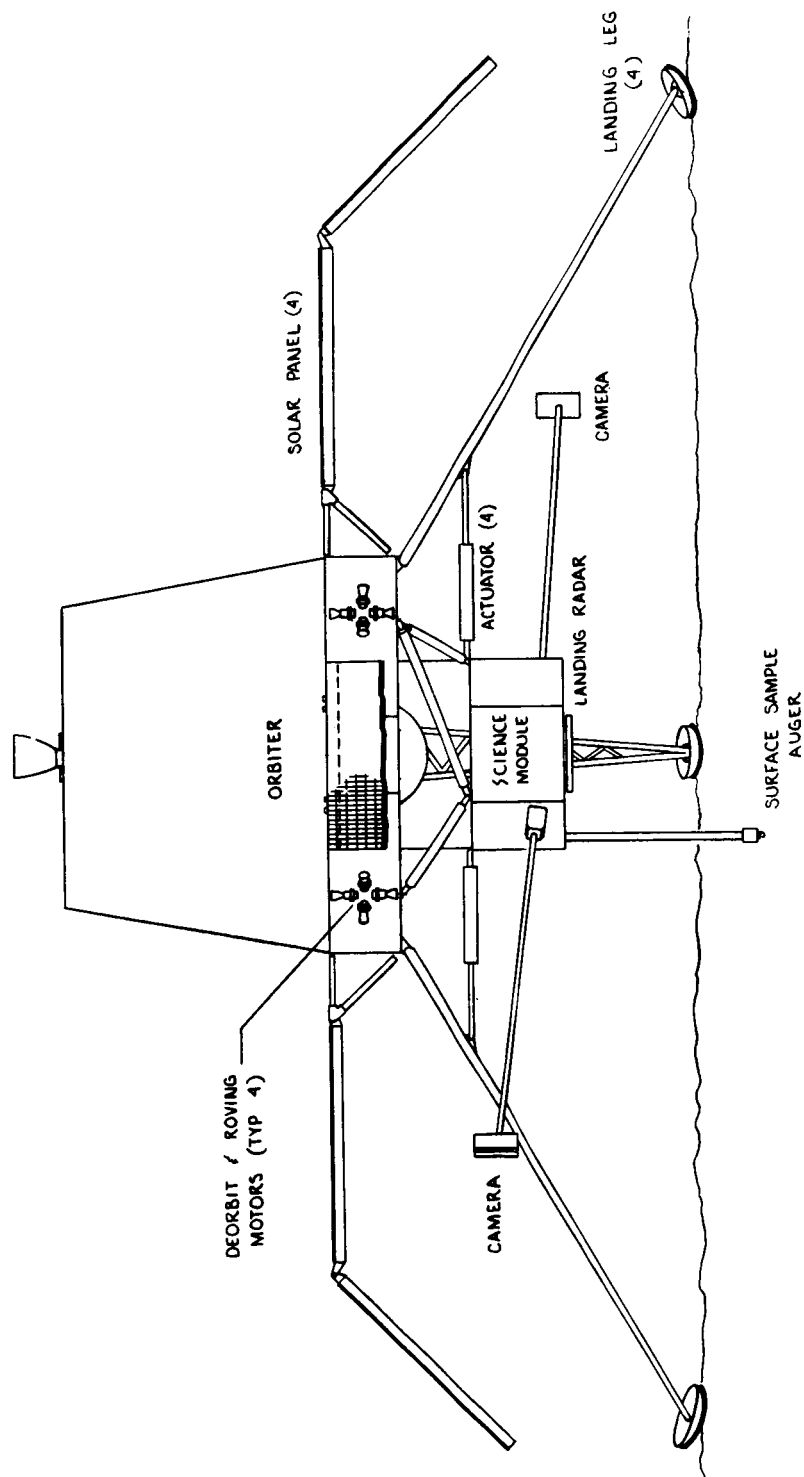


Figure V-6 Phobos/Deimos Landed Orbiter (Landed Configuration)

C. RENDEZVOUS AND LANDING MISSION COST ESTIMATE

Program cost estimates for the separable lander and the landed orbiter versions of the baseline rendezvous and landing mission were developed. Because later mission baselines, i.e., sample return and combination Mars and satellite missions, would require large payload capabilities, the landed orbiter concept was used for cost baselining. The summary in Table V-2 indicates the estimated cost of a two-launch mission in the 1979 opportunity, using Titan IIIE/Centaur launch vehicles. The costs are expressed in equivalent FY '72 dollars.

Table V-2 Phase I Cost Summary - Phobos Landing

	(\$ in Millions)				
Spacecraft (Landed Orbiter)					198
Science					30
Other NASA Costs					52
					<u>280</u>
Launch Vehicles (2)					44
					<u>324</u>
TOTAL					<u><u>324</u></u>
Funding by Fiscal Year					
<u>1976</u>	<u>1977</u>	<u>1978</u>	<u>1979</u>	<u>1980</u>	<u>1981</u>
18	62	82	69	38	29
					<u>1982</u>
					26
					<u>Total</u>
					324

Phase II - Satellite Sample Return Mission

VI. PHASE II - SATELLITE SAMPLE RETURN MISSION

In Phase I an understanding of the satellite rendezvous and landing mission was developed and a measure of the feasibility of performing it was determined. Our instructions from the Technical representative of the Contracting Officer for Phase II were to define a mission profile and a spacecraft configuration for returning a surface sample of Phobos or Deimos back to Earth for detailed analysis. To accomplish this objective it was necessary first of all, to compile and evaluate the scientific rationale for such a mission.

The answer to the question, "Why a Phobos/Deimos sample return mission?" is, "Because we can learn more from Earth laboratory work." The experimental techniques required for accurate age dating, exact elemental and mineralogical analysis, detailed studies of physical and chemical properties, and reading the history of bombardment, impact and exposure, cannot be performed remotely. The kinds of instrumental techniques needed for this work include:

- Solid source mass spectrometry;
- Neutron activation analysis;
- Electron microprobe;
- Electron microscope;
- Petrographic studies;
- Natural radioactivity;
- Magnetic and electrical properties;
- Particle tracks;
- Mössbauer spectroscopy;
- Thermal properties;
- Evolved gas analysis.

A. SAMPLE RETURN MISSION PROFILE

For the baseline sample return mission, the launch opportunity was changed from 1979 to 1981 to allow more time for the additional mission design and hardware development that would be required. This change in launch year necessitated rerunning the Phase I performance calculations, but the basic profile from Earth launch to satellite landing remained the same. The mission profile from the surface of Phobos back to Earth is shown in Figure VI-1. A small velocity is required to escape the satellite. Thereafter, the return spacecraft orbit about Mars is changed to a 1500 Km by 95000 Km altitude ellipse and a plane-change maneuver is performed to line up with the required trans-Earth trajectory. The principal delta velocity requirements for this sequence are:

Raise apoapsis to 95,000 Km - 746 mps

Lower periapsis and plane change - 110 mps

Trans-Earth injection - 712 mps (1983/84 Mars launch)

The total velocity budget including miscellaneous corrections and allowances is 1786 mps.

It might be appropriate at this point to insert the results of some interesting work done by our contract technical monitor, Mr. Harrison, on the comparison of energy requirements for a number of sample return missions in the solar system. Figure VI-2 compares total ΔV requirements for sample return missions to Phobos (more difficult than Deimos), our Moon, Eros (one of the easiest asteroid missions) and Mars. It is rather surprising to see that the Mars satellite missions are less demanding than even our own Moon. This is due to the difference in the required ascent velocity from the surface, the Moon's larger gravity necessitating a much higher ΔV for this maneuver.

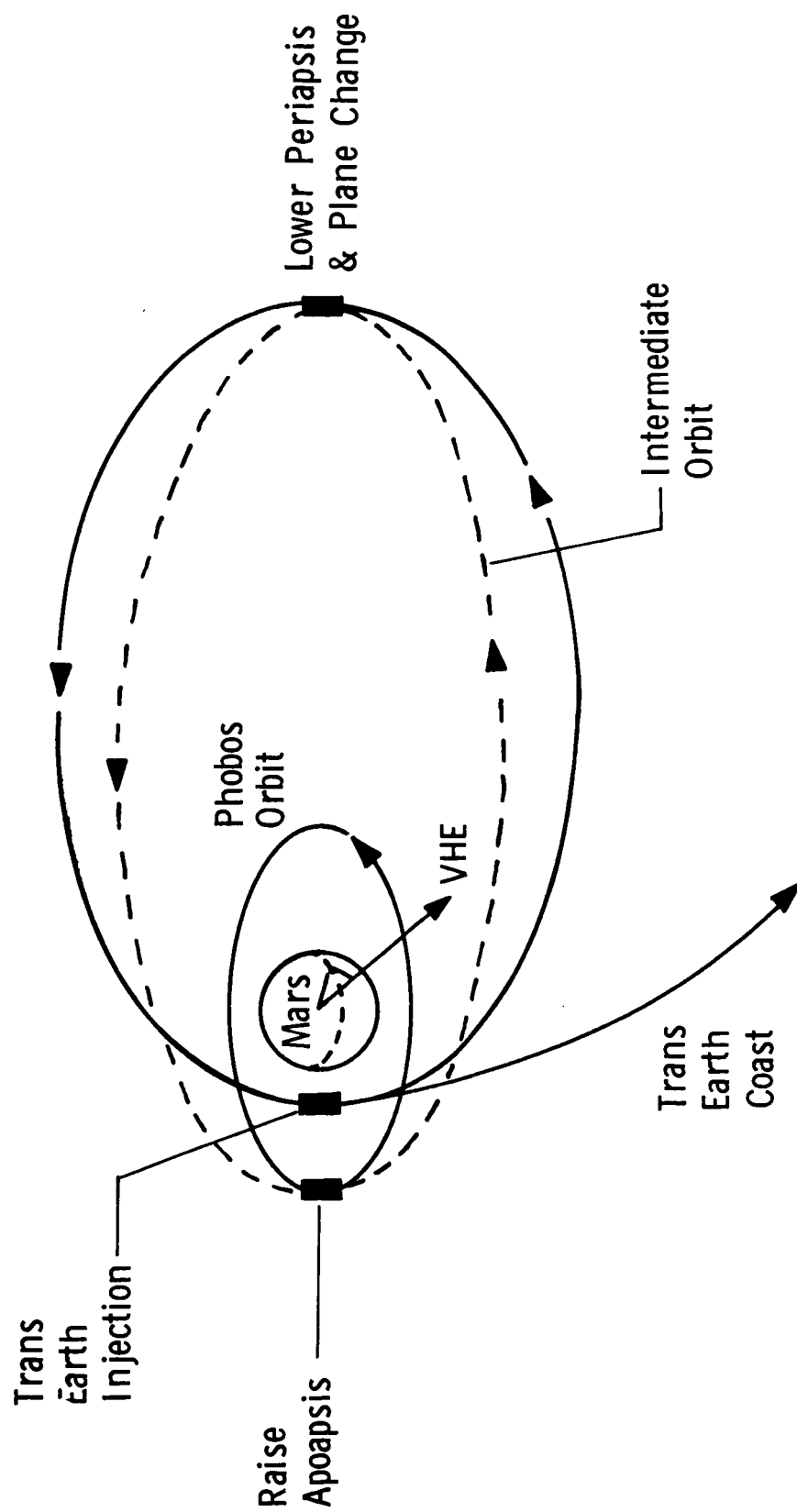
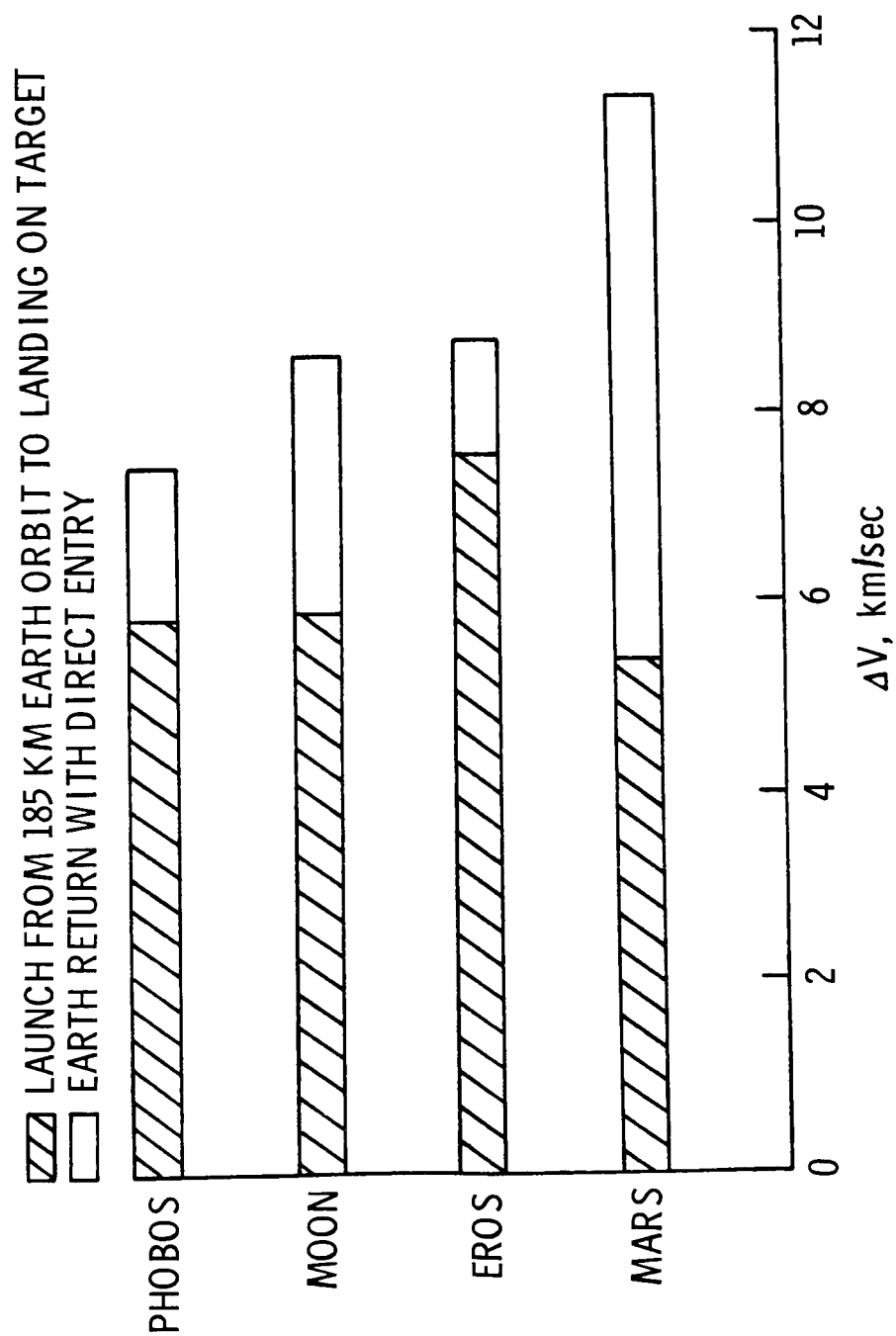


Figure VI-1 Overview of Mission Profile - Sample Return

Figure VI-2 Comparative ΔV Requirements for Sample Return Missions

B. SAMPLE RETURN MISSION BASELINE SPACECRAFT

A rather extensive systems trade study was conducted in Phase II to select a recommended space vehicle system concept for a Martian satellite sample return mission. The mission configuration elements shown in Table VI-1 were assembled in various combinations and comparatively analyzed to establish the baseline.

The baseline selected is shown in Figure VI-3. It consists of a modified Viking Orbiter adapted to the landed Orbiter configuration. The Orbiter delivers a modified Venus Pioneer (Planetary Explorer) Earth return vehicle to the satellite surface.

Modifications to the Planetary Explorer are necessary to achieve proper dynamic balance when the liquid propulsion system and Earth Entry Module are incorporated for sample return. The same communications and guidance subsystems are employed. The power subsystem does not require a battery; series-parallel cell switching is employed to maintain a desired voltage output. The thermal control system is modified for heat retention instead of heat rejection.

The Viking orbiter is modified to incorporate landing legs, landing radar, sampling subsystem, the earth return vehicle, and a 38% stretched propulsion system. Thermal control systems are modified for the landed mission. Solar panels are integrated with the landing legs.

The total weight of this spacecraft is 3374 Kg (7439 lbs) which includes an allocation of 260 Kg (570 lbs) for the Earth return vehicle and its associated equipment.

The operational sequence of the sample return is shown in simplified form in Figure VI-4.

Table VI-1 Candidate Sample Return Mission Elements

Launch Systems:	Titan IIID/Centaur
	Titan IIIC
	Titan IIID(7)/Centaur
	Shuttle/Centaur
Satellite Delivery Systems:	38% Stretch Viking Orbiter
	60% Stretch Viking Orbiter
	Staged Viking Orbiter
	Space Storable Propellant Viking Orbiter
	Propulsion Module with Round-Trip Control Module
Satellite Landing:	Landed Orbiter
	Separable Lander
Earth Return Vehicles:	Modified Planetary Explorer
	New 3-Axis Stabilized Vehicle
	Mariner Derivative
	Round Trip Control Module

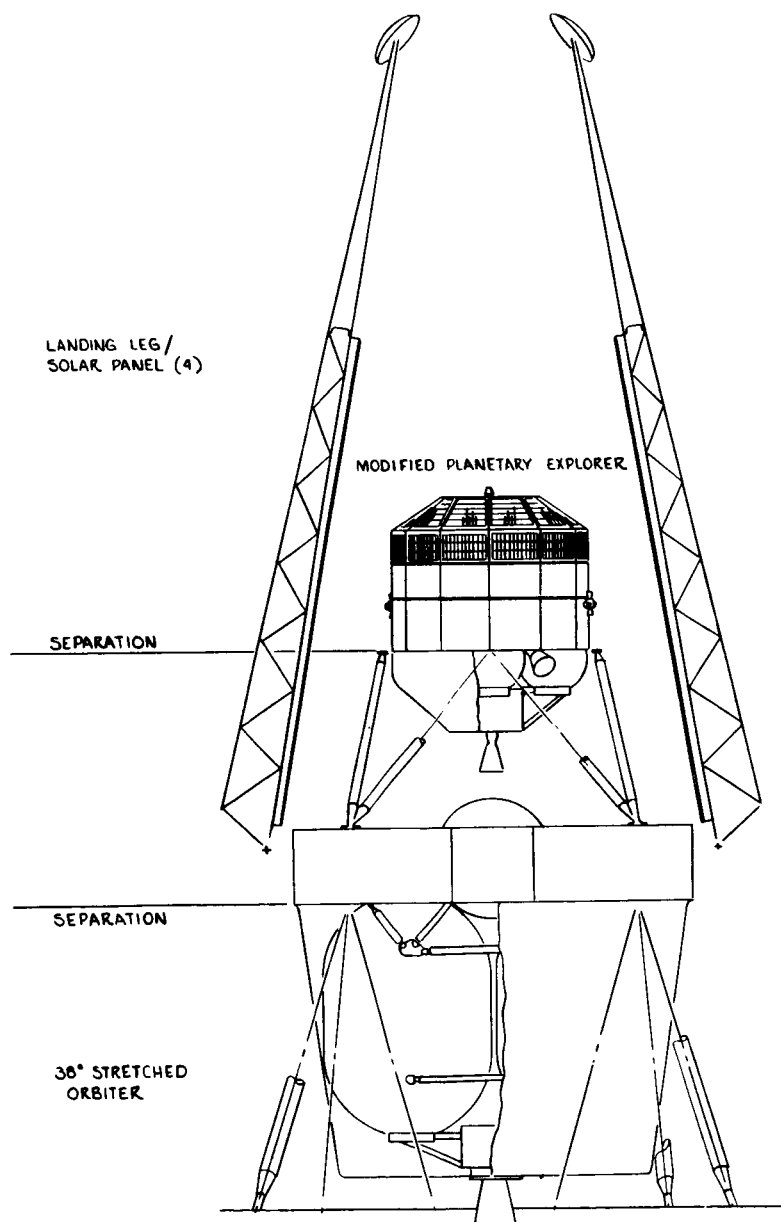


Figure VI-3 Baseline Sample Return Configuration

VI-8

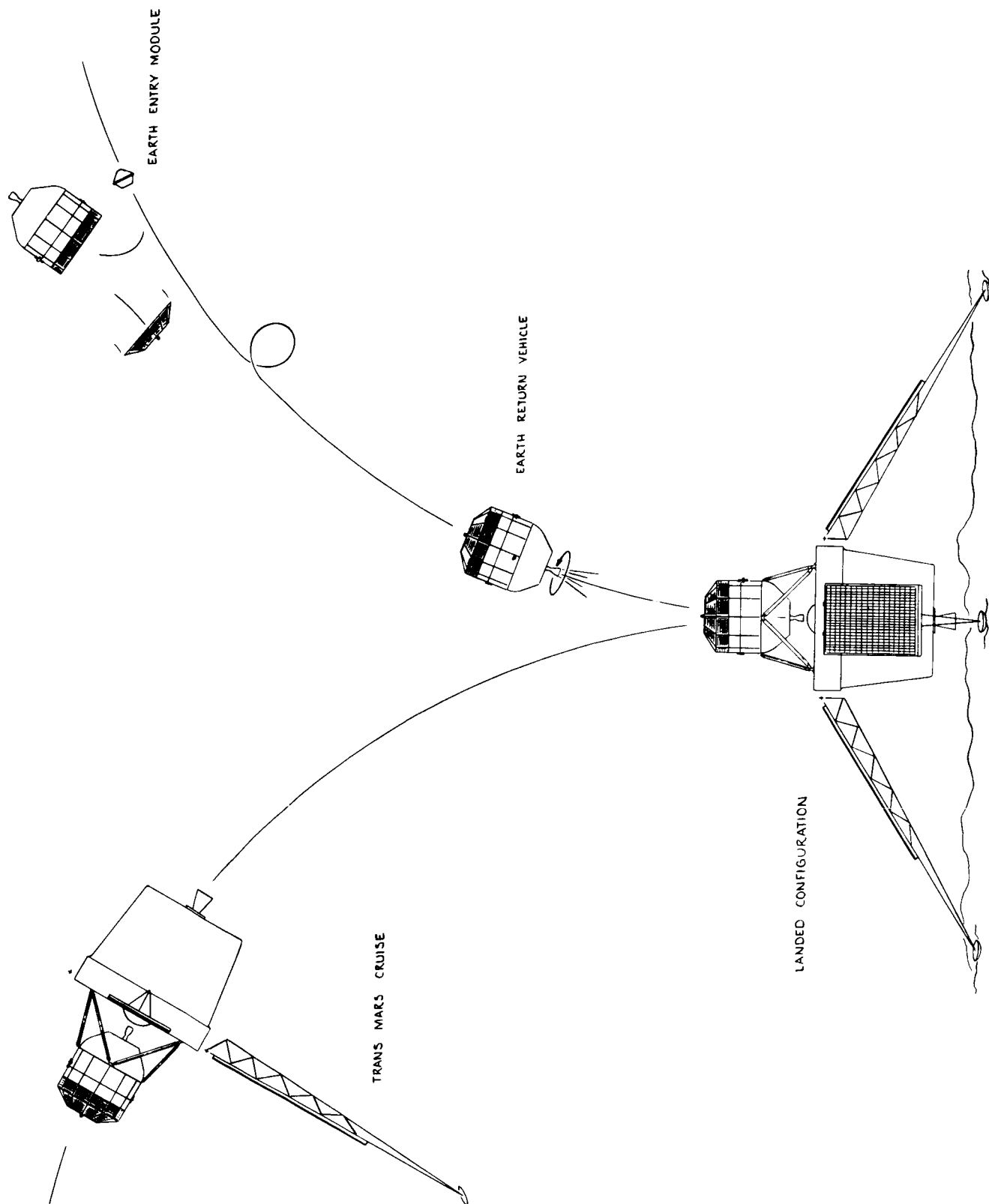


Figure VI-4 Baseline Sample Return Sequence

Following touch-down, Earth communications are established. Sampling is accomplished under control of the orbiter CC&S with Earth-based supervision. The earth return vehicle and orbiter remain on the surface and Earth return alignment data is deduced. Earth return vehicle liftoff is programmed to provide the best alignment for initial Earth communications to the return vehicle. The modified Venus Pioneer vehicle uses the spinning celestial attitude determination system (SCADS) guidance scheme for the Mars orbital and interplanetary cruise phases.

An alternate spacecraft configuration that was studied in some detail in Phase II is shown in Figure VI-5. It consists of an integrated three-axis stabilized control module/lander with a propulsion module which is jettisoned prior to final closure and touch-down. The subsystems contained in the earth return vehicle function throughout the mission so that there is no duplication of hardware. The injected weight of the configuration is 2500 kg compared with 3375 kg for the baseline concept. The earth return vehicle (control module) is a new light-weight three-axis stabilized vehicle composed of proven or currently identified interplanetary spacecraft subsystems. The integrated lander module evolved from the Phase I lander/rover, houses the communications (20 watt TWTA's), primary power system (RTG and batteries), sampling subsystem, landing radar, tape recorders, and approach navigation TV cameras. The articulated 30" parabolic high gain antenna is located on the earth return vehicle. The propulsion module which is jettisoned prior to final closure and touch-down is the basic Viking '75 propulsion system (no stretch required). Terminal descent propulsion is a hydrazine system.

The sample return earth entry module shown in Figure VI-6, is designed to survive the loads imposed during Earth entry

VI-10

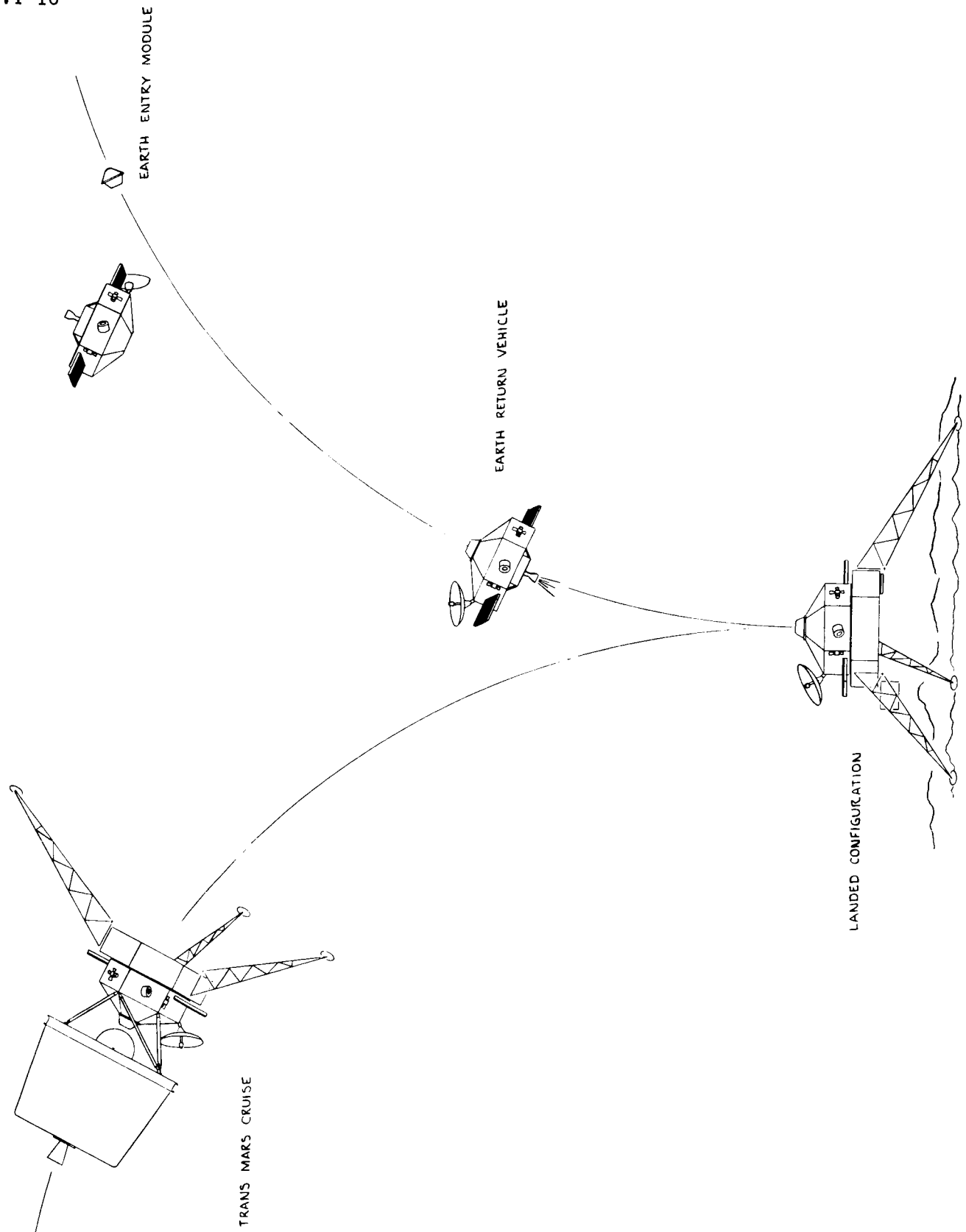


Figure VI-5 Preferred Alternate Sample Return Sequence

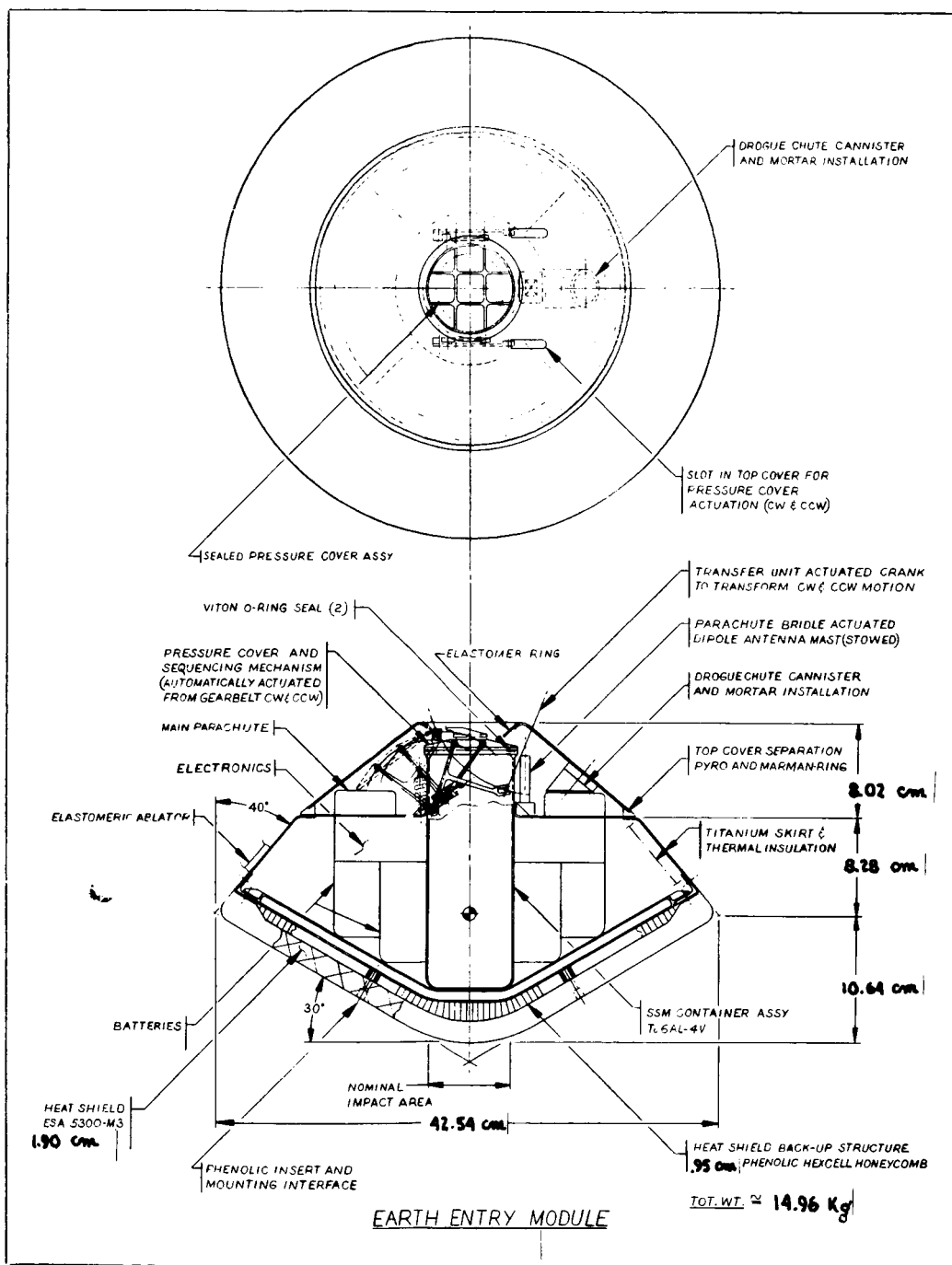


Figure VI-6 Earth Entry Module

and impact. The capsule design is based on Apollo technology and can accommodate entry velocities up to 12.8 km/sec. A parachute is deployed as soon as the velocity is subsonic. Electronic tracking aids are provided to accomplish recovery of the capsule. This configuration provides for the stowage, sealing and environmental protection of a 2 kg Phobos surface sample.

C. SAMPLE RETURN MISSION COST ESTIMATE

The cost summary for the baseline sample return mission in FY'72 dollars (no escalation factors added) is shown in Table VI-2. The cost estimate has been built up using a work breakdown structure patterned after the Viking '75 Lander system. This WBS contains over 80 elements of cost. Labor and material estimates were made for each of the WBS elements. Two previously developed program estimates were used as references and calibrations for this estimate: 1) the Viking '75 program (which should have higher costs for equivalent elements because of the completely new developmental nature of the work); and 2) the Viking '77 program (which should be lower for equivalent elements because it involves minimum modification to existing designs). This estimate is based on two launches in 1981 using Titan IIIE/Centaur launch vehicles.

Table VI-2 Sample Return Cost Summary

	(\$ in Millions)				
Spacecraft (Landed Orbiter)	192				
Return Vehicle (Modified PE/Pioneer)	96				
Science (Includes Sample Collection)	35				
Other NASA Costs	79				
	<hr/>				
	402				
Launch Vehicles	44				
	<hr/>				
TOTAL	<u>446</u>				
	<hr/>				
	Funding by Fiscal Year				
	<hr/>				
	1977	1978	1979	1980	1981
		<hr/>	<hr/>	<hr/>	<hr/>
	25	86	112	94	53
				40	36
				<hr/>	<hr/>
				1982	1983
				<hr/>	<hr/>
				Total	446

Phase III - Combined Mars and Phobos/Deimos Missions

VII. PHASE III - COMBINED MARS AND PHOBOS/DEIMOS MISSIONS

The final three months of the study was directed at defining missions that combine landings on Mars and Phobos/Deimos exploration within one mission profile. The objective was to determine whether or not a properly designed combined mission can meet both Mars and satellite science objectives at a program cost substantially less than doing the two missions separately.

The major challenge in Phase III was to define and compare the large number of possible combined mission configurations to arrive at a recommended baseline.

The science objectives assumed for the Mars mission were focused on the geosciences: geology, geophysics and geochemistry. This is consistent with the general recommendations that are being made by the science community for post Viking '75 Mars exploration. Science requirements for Phobos/Deimos exploration varied according to the mission modes that could be performed in different combination event profiles. Satellite mission modes considered were observation orbit, station-keeping and landing. Figure VII-1 shows the baseline science payloads that were used with the candidate combined missions.

A. COMBINED MISSION PROFILE

The basic elements of the recommended baseline combined mission profile are shown in Figure VII-2. The major decisions and trades that had to be analyzed to arrive at this baseline were:

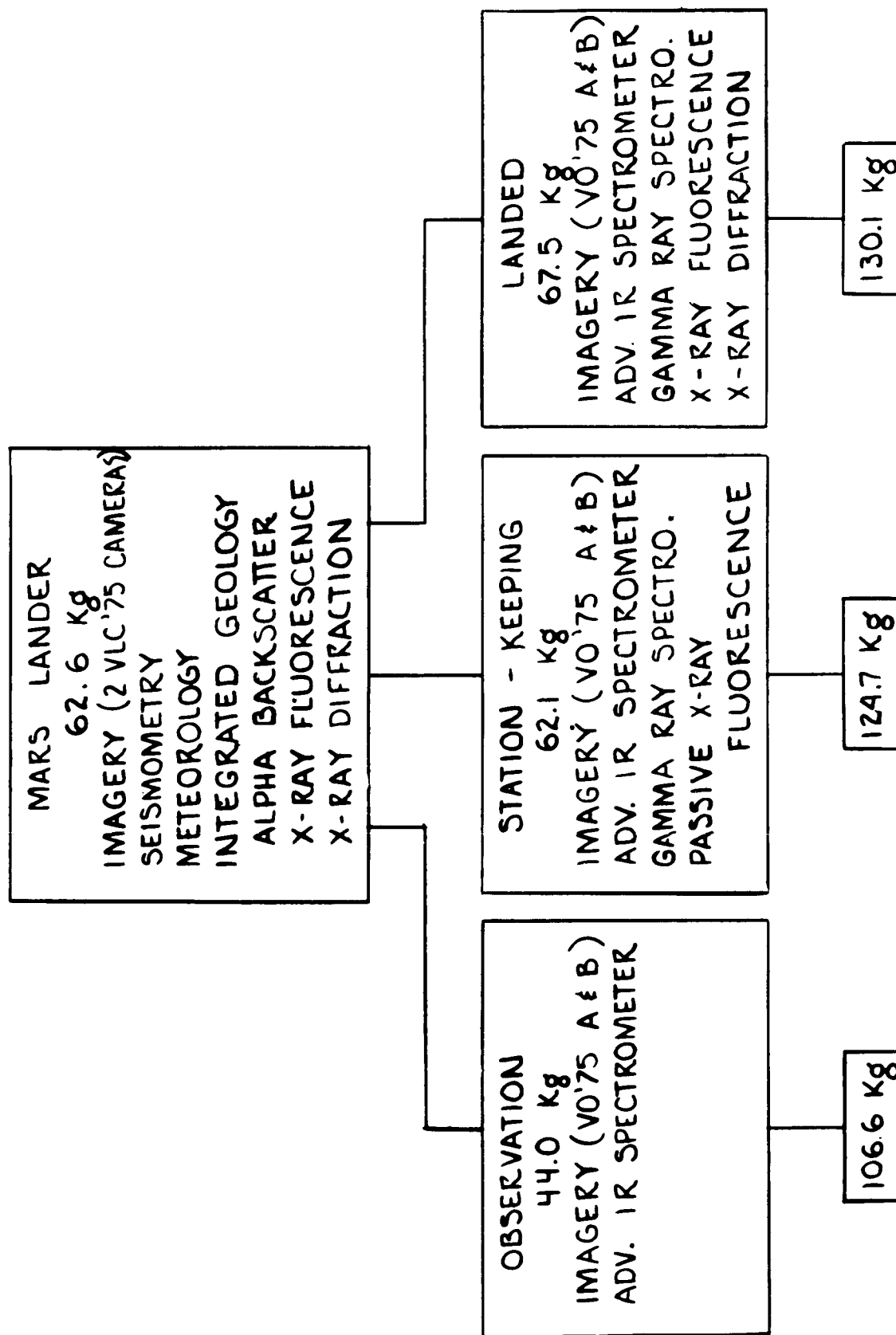


Figure VII-1 Nominal Science Payload Allocation

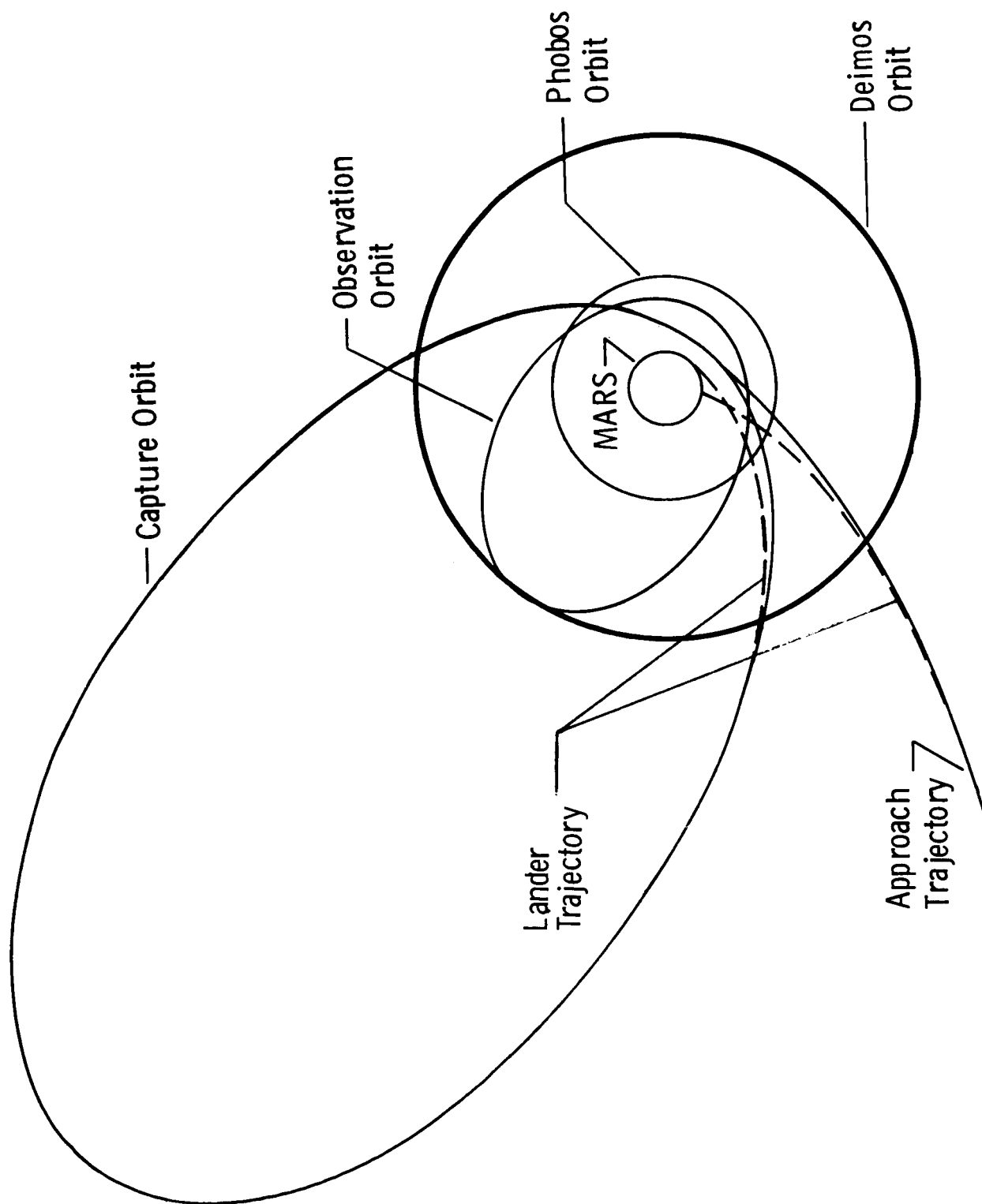


Figure VII-2 Overview of Mission Profile

- 1) Selection of capture orbit period;
- 2) Mars landing mode (direct entry or out-of-orbit)
- 3) Mars landing latitude accessibility;
- 4) Selection of observation orbit (15 or 30 hour period; synchchronized with Phobos or Deimos);
- 5) Spacecraft (stretched, staged or high energy propulsion systems).

The 97-hour capture orbit was selected because it minimizes the Mars orbit insertion propellant requirements while still keeping out-of-orbit entry velocities reasonably low. Mars landing out-of-orbit was selected to allow mission flexibility and to minimize the modifications required to the Viking '75 Lander design.

The easiest combined mission to perform are those in which Mars landing occurs near the equator. Increased Mars landing latitude accessibility can be achieved only by increasing the spacecraft propulsion capability or by reducing the payload delivered to the satellite. Our baseline mission allows for Mars landings generally in the range of $\pm 12^\circ$ in latitude about the equator.

The 15-hour observation orbit, synchronized such that it meets Deimos on every other revolution, turned out to be the most desirable choice. It provides for more and closer observations of the second satellite than other observation orbits.

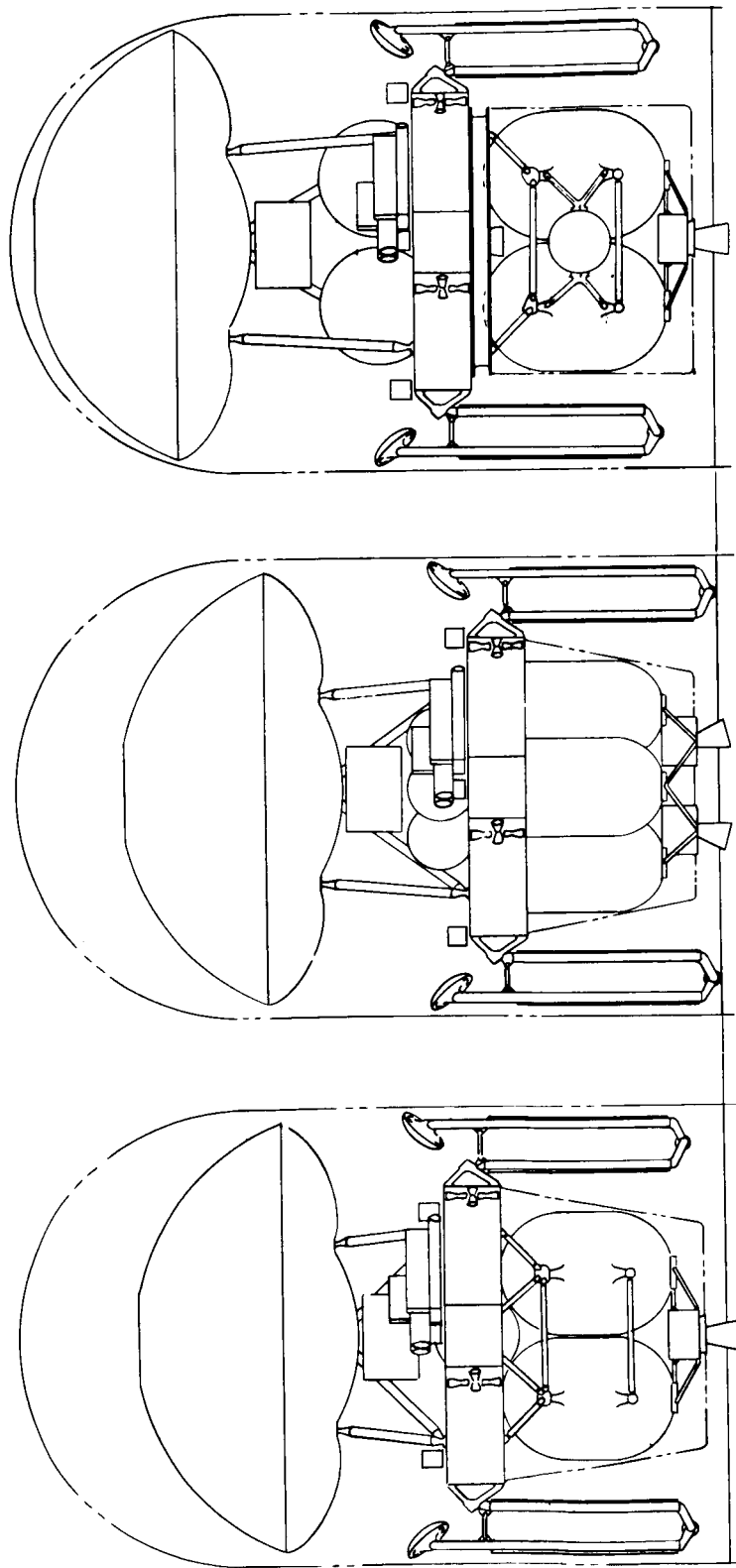
The stretched Viking '75 Orbiter spacecraft was the choice for the baseline because its performance capabilities met the 1979 baseline launch year requirements and it necessitates less orbiter modification than other candidates.

B. COMBINED MISSION BASELINE SPACECRAFT

In selecting the baseline spacecraft configuration for the recommended combined mission, we examined three basic concepts: the stretched Viking Orbiter two tank and four tank versions); the staged orbiter; and the high energy space storable propellant orbiter. These configurations are shown in Figure VII-3. The high energy propellant orbiter would be a derivation of the stretched orbiter.

When these spacecraft configurations are combined with the alternative mission modes studied, a total of 324 possible combined missions result. Performance data is available from our Phase III results for all of them. Figure VII-4 is a typical performance curve for a stretched orbiter, 1979 launch, and Mars landing out of a 97-hour orbit. It shows that our baseline mission (Phobos landing with a 67 Kg payload) can be performed with a Titan IIIE/Centaur launch vehicle.

Figure VII-5 outlines the recommended baseline spacecraft and indicates the changes required from the Viking '75 Lander/Orbiter. The total injected weight of this vehicle is 4150 Kg (9150 lbs) compared with the Viking '75 injected weight of approximately 3800 Kg. Both of these injected weight figures include a 210 Kg allowance for spacecraft adapter, launch vehicle peculiar equipment, and project reserve. The Titan IIIE/Centaur can inject an equivalent 4157 Kg to Mars over a 30-day launch period in the 1979 launch opportunity.



STAGED ORBITER

FOUR TANK ORBITER

BASELINE CONFIGURATION

- o STAGE I
38% PROPELLANT INCREASE OVER '75
- o STAGE II
50% OF STAGE I PROPELLANT LOAD

- o 100% PROPELLANT INCREASE OVER '75

- o 26% PROPELLANT INCREASE OVER '75

Figure VII-3 Combined Missions Spacecraft Configurations

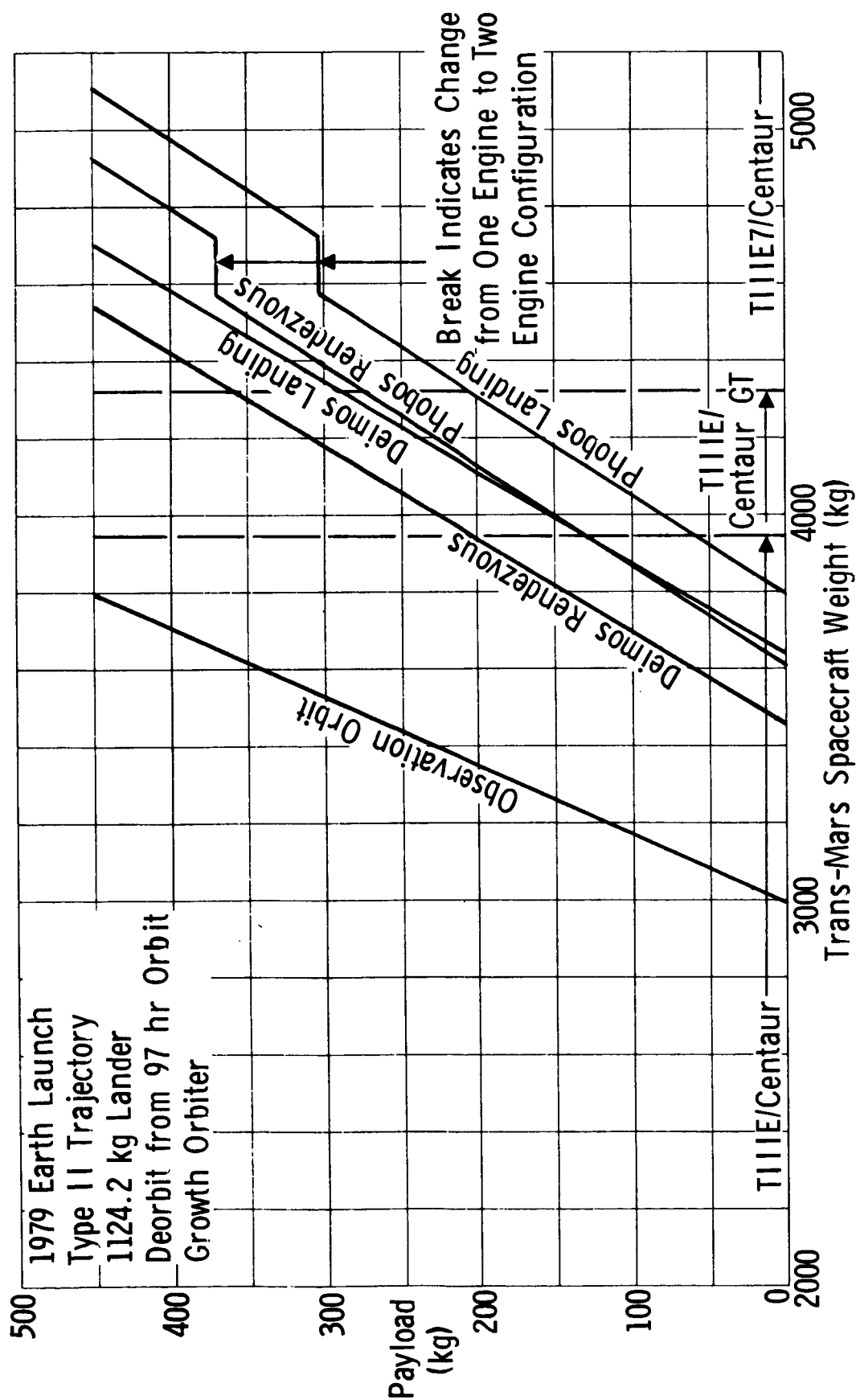


Figure VII-4 Trans-Mars Spacecraft Weight vs Payload - 1979 - 97 Hour Orbit

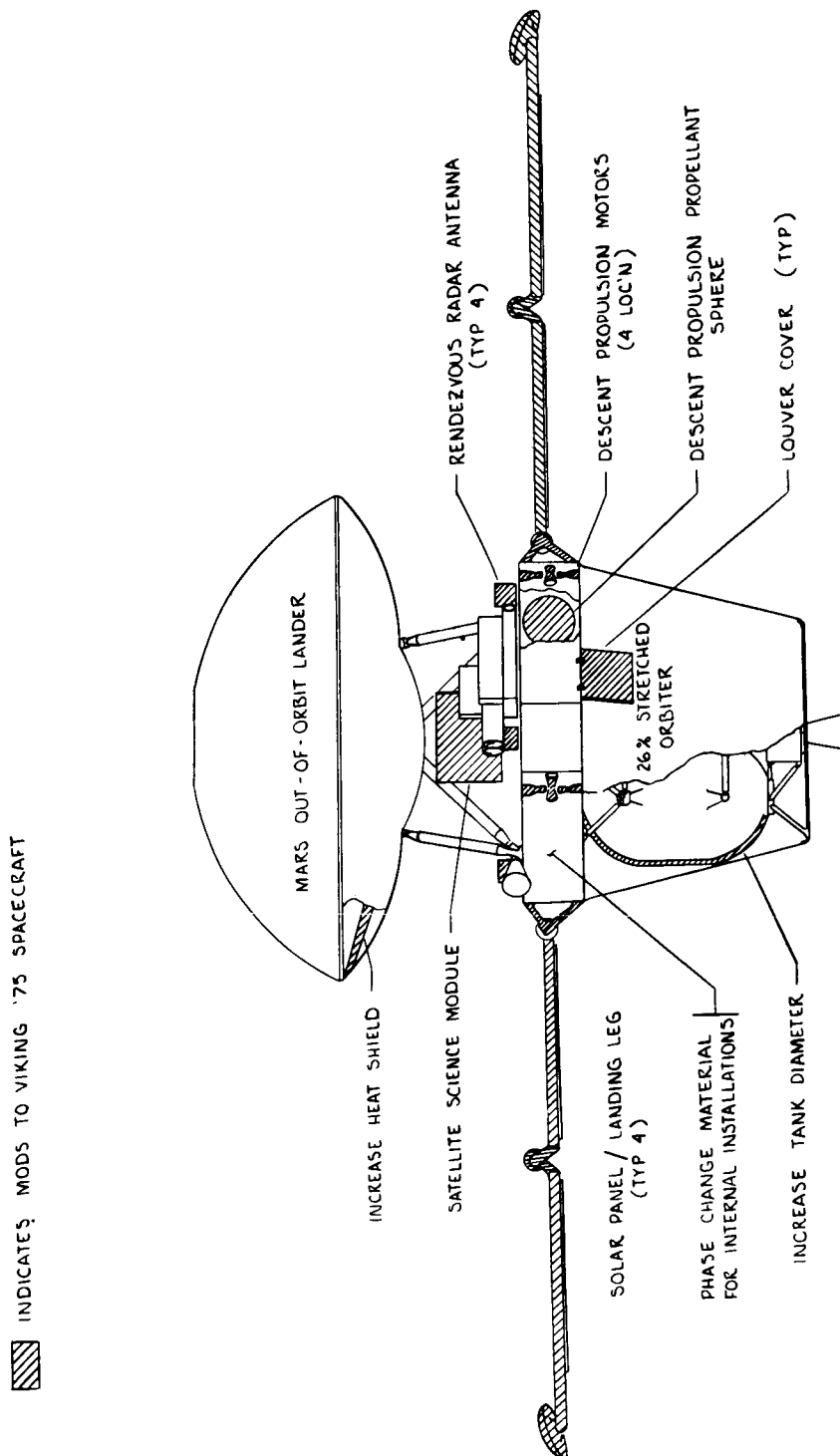


Figure VII-5 Combined Missions Baseline Spacecraft (Cruise Configuration)

C. COMBINED MISSION COST ESTIMATE

The estimated cost of the recommended baseline combined mission is summarized in Table VII-1 in FY '72 dollars. This covers the Mars landing out of a 97-hour orbit and a Phobos rendezvous and landing using the landed orbiter configuration. This estimate assumes that the total program would be managed by one contractor which allows some sharing of functions and costs between the Mars lander and the landed orbiter. The estimate is for two launches using the Titan IIIE/Centaur.

Table VII-1 Combined Mission Cost Summary

(\$ in Millions)							
Spacecraft (Landed Orbiter)							
						179	
Mars Lander (Out of 97 Hr Orbit)							
						95	
Science							
						45	
Other NASA Costs							
						78	
						397	
Launch Vehicles							
						44	
TOTAL							
						441	
<u>Funding by Fiscal Year</u>							

Conclusions

VIII. CONCLUSIONS

The results of this study have substantiated the original Pritchard and Harrison work in demonstrating that a number of mission concepts to explore Phobos and Deimos are feasible and practicable. The study demonstrated further that systems, subsystems and operational modes could be derived directly from Viking '75 and other current programs to perform the Mars satellite missions, thereby minimizing costs and maximizing mission success potential.

In surveying the scientific community to measure interest in Phobos/Deimos exploration, we encountered an unusual mixture of feelings. While we found no active and forceful champions of a Mars satellite mission, we repeatedly found easily excited curiosity and conjecturing among space scientists about the origin and nature of these tiny bodies. This undercurrent of scientific interest, which has been given impetus by the recent returns of Mariner 9, may be the forerunner of well defined and enthusiastically supported recommendations for exploring the moons of Mars. If this is the case, NASA's decision to conduct this study may prove to be a very timely one.

Out of all the mission profiles studied in this work, the type that appeared to be most attractive were those that combined Mars and Phobos/Deimos exploration. A comparison of estimated costs for a number of Mars satellite missions, Mars landing missions, and combinations of the two, provides an interesting measure of the cost effectiveness of the combined mission approach. The cost estimates shown below, all normalized to a 1979 Mars lander/orbiter mission designed and managed to the same groundrules used in this study, illustrate the point:

<u>Mission</u>	<u>Relative Program Cost</u>
Mars Lander/Orbiter	1.0
Phobos/Deimos Landing	0.8
Phobos/Deimos Sample Return	1.14
Mars + Phobos/Deimos Observation Orbit	1.04
Mars + Phobos/Deimos Rendezvous	1.07
Mars + Phobos/Deimos Landing	1.14
Mars + Phobos/Deimos Sample Return	1.47

As these relationships indicate, satellite missions can be added to Mars mission profiles at very little increase in total program cost. When one considers that the well-recognized science objectives and requirements for Mars exploration plus the puzzling scientific mysteries of the satellites can all be addressed through such combined mission profiles, a very attractive return on investment begins to emerge.

In the course of this study we identified a number of technology areas where further study and research would contribute to Phobos/Deimos mission planning and development. They include the following:

- 1) For all satellite missions:
 - a) navigation analyses for automatic orbital maneuvers and rendezvous;
 - b) lightweight, low-power rendezvous radar systems;
 - c) satellite sampling and tie-down techniques;
 - d) landed orbiter thermal control, power, propulsion and structural analyses;
 - e) mobility and navigation mechanization for satellite rovers;

- f) adaptive science payloads for satellite exploration;
 - g) universal space storable propellant propulsion module.
- 2) For satellite sample return missions:
- a) lightweight, low-power 3-axis and closed-loop spin stabilized guidance and control subsystems;
 - b) automatic spacecraft alignment and ascent guidance subsystems;
 - c) sample collection, stowing and protection techniques.
- 3) For combined missions:
- a) increased Mars latitude accessibility;
 - b) improved Mars landing accuracy
 - c) Mars entry corridor analysis
 - d) mapping profiles and sensors for combined missions;
 - e) small instrumented probes for combined missions.